NACA

RESEARCH MEMORANDUM

PRELIMINARY ANALYSIS OF EFFECTS OF AIR COOLING TURBINE BLADES ON TURBOJET-ENGINE PERFORMANCE

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

The effects of turbine-blade cooling on engine performance were analytically investigated for a turbojet engine in which cooling air is bled from the engine compressor. The analysis was based on the measured performance and operating conditions of a typical uncooled axial-flow turbojet engine in order to obtain a fixed basis for comparison between operation with two cooled-blade configurations and the performance of the uncooled engine.

The calculations were made for a constant turbine-inlet temperature, engine speed, and flight Mach number over a range of altitudes. The analysis considered the effects of heat transfer on coolant requirements, blade-coolant-passage pressure drop, and turbine performance as well as the effects of heat loss, compressor bleed, coolant pumping requirements, and other factors on engine performance. As a part of the analysis, the minimum cooling requirements permitting substitution of nonstrategic metals in turbine blading and the desirable characteristics of high-temperature turbojet engines were also considered.

The results indicated that, for a constant turbine-inlet temperature and engine speed, air cooling of the turbine blades increased the specific fuel consumption and decreased the thrust of the engine. For a given coolant flow, the percentage increase in specific fuel consumption was less than the percentage decrease in thrust, relative to the uncooled engine. The required coolant-flow ratio, defined as the ratio of coolant weight flow to compressor-air weight flow, increased with altitude. Because of the high coolant-supply pressure required, it was generally necessary to bleed the cooling air from the compressor.

The highest possible cooling effectiveness was desirable to minimize the coolant weight flow and its effects on engine performance.

The analysis indicated that appreciable reduction in blade-alloy content may be feasible with some sacrifice in over-all engine performance. The application of cooled turbines to permit operation at high turbine-inlet temperature with appropriate modifications in design of other components offers possibilities of improvement in performance at both maximum-power and maximum-range conditions and improved flexibility in engine and aircraft operation. 1331

INTRODUCTION

Theoretical studies of turbojet-engine cycles indicate that substantial gains in aircraft performance can be obtained with engine operation at gas temperatures that are beyond the temperature limitations of current uncooled turbines. It is evident that the high-temperature turbine requires cooling to permit operation with currently available materials, although the engine performance is diminished by the effects of cooling losses. Another use of cooled turbines is to provide satisfactory endurance at current gas temperatures, with low-temperature materials having a minimum content of strategic alloying elements that could become inaccessible in an emergency.

The previous analyses of turbine cooling, such as the blade-cooling studies reported in reference 1, were made for the purpose of determining which blade configurations had the necessary cooling capacity to offer possibility of substantial increases in gas temperature or reduction in strategic-metals usage. In reference 1 it is recognized that the cooling losses might alter the relative comparison of configurations and the extent of desirable increase in gas temperature or reduction in use of strategic metals.

An analysis of the effects of air cooling on engine performance at current gas temperature was made at the NACA Lewis laboratory. The purpose of this report is to present results illustrating the relative magnitudes of the effects of air cooling for two blade configurations and several possible blade metals. The future potential performance and design requirements of improved air-cooled turbojet engines are also considered.

In order to make a detailed analysis of these effects, knowledge of the turbine geometry and operating conditions is essential in evaluating heat-transfer characteristics under various flight conditions. Consequently, most of the analysis presented in this report considers only application of an air-cooling system to a

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typical turbojet engine, for which the uncooled performance and the operating conditions are known from engine tests. The use of a given engine as a basis for the analysis has distinct advantages in comparison with an ordinary cycle analysis because it permits study of altitude effects and the effects of cooling on compressor and turbine operation. This method also provides a fixed basis for comparison between the cooled and the uncooled engine and is generally more realistic than the simplified cycle analysis.

Some assumptions, which affect the significance of the final results, are required in the cooling analysis although the methods include most of the variables. With any specified allowable blade-metal temperature distribution, the analysis provides good comparative evaluation of air-cooling systems. The greatest uncertainties are found in the specification of the allowable metal-temperature distribution in the blade as a function of the alloy content and physical properties of the metal, and the behavior of the metal under the uncertain temperature and stress conditions prevailing in the actual engine.

ANALYSIS

The proposed turbine configuration that has been used as a basis for the present analysis is illustrated in figure 1. The figure shows a half section of the turbine rotor and casing, and part of the jet-nozzle assembly. The combustion gases are introduced from the left side of the figure into the turbine stator and rotor blading and subsequently pass downstream into the jet nozzle. The rotor-blade cooling air, bled from the compressor, is introduced through a supply pipe to the stationary diffusing section at the turbine hub and passes into a vaned shroud (on the downstream face of the wheel), which carries the air to the blade base. The cooling air then passes through the hollow rotor blade and is discharged at the tip to mix with the main flow of working fluid. In this analysis, the stator blades are also considered to be cooled with compressor air introduced through an external manifold. The air passes down the leading portion of the stator blade and is discharged from small holes distributed over the trailing portion of the blade to achieve a film-cooling effect.

Two types of rotor blade are considered in the analysis, both of which incorporate special modifications to improve control of blade temperature. The two blade-coolant-passage configurations are illustrated in figure 2.

Outline of Analysis

The outline of the analysis logically falls into three main steps: First, the coolant flow required at any given set of conditions is established; second, the flow conditions of pressure and temperature through the blade passage and rotor supply shroud are determined in order to establish the required supply pressure and temperature of the air at the hub of the turbine rotor; and finally, the ultimate effect of the cooling on jet-nozzle conditions, engine thrust, and specific fuel consumption are computed. These steps are considered in greater detail in the following sections.

Determination of coolant-flow requirements. - The blade-metal temperature distribution is primarily dependent on the radial gas-temperature distribution, the coolant-inlet temperature at the blade root, the blade-profile heat-transfer coefficient, the coolant-passage heat-transfer coefficient, and the configuration of the coolant passage. These variables can be expressed in equations within limitations that are briefly discussed later in the report. In this analysis, the coolant temperature at the blade root is arbitrarily specified at a constant value for all calculations inasmuch as the air temperature at the point of compressor bleed is unknown in the first stage of the analysis. It is therefore assumed that the temperature of the cooling air supplied at the turbine hub can be controlled by a heat exchanger so that the specified temperature at the blade root is obtained. The practical necessity of providing this air-temperature control is subsequently discussed. In determining the required coolant flow, the known engine operating conditions for any selected flight condition are used to evaluate the blade Reynolds number and the outside blade-profile heat-transfer coefficient. For a specified coolant-passage configuration and allowable metal temperature distribution, it is possible to determine the weight flow of coolant required, at the specified turbine-inlet gas temperature and the specified inlet coolant temperature at the blade root, with consideration of the effects of heat transfer and centrifugal compression on the cooling-air temperature as it passes through the blade. Essentially the process is a trail-and-error solution for the passage heat-transfer coefficient that yields the desired metal-temperature distribution. The rotor coolant weight flow for any set of conditions can then be nondimensionally expressed by dividing by the total compressor-air weight flow that was initially used in evaluating the blade Reynolds number and heat-transfer coefficient. This ratio, designated the coolant-flow ratio, can then be evaluated for a range of flight conditions inasmuch as the variations of engine-gas weight flow and compressor pressure ratio, which affect the heat-transfer coefficients, are known from tests.

The stator-blade coolant-flow requirements for this particular analysis were estimated from limited data for film-cooled blades presented in reference 2. A fixed stator coolant-flow ratio of 0.01 appeared adequate for all conditions. No allowance was made for cooling of such components as the turbine casing and jet-nozzle assembly.

1331 Analysis of flow conditions in cooling passage. - After the coolant-flow requirements for the rotor blade have been evaluated, the flow conditions in the cooling passage can be analyzed to determine the Mach number distribution in the passage and, subsequently, the required pressure at the blade root. The analysis of Mach number distribution is made according to methods given in reference 3, and takes into account the effects of centrifugal compression, friction losses, momentum change due to heat transfer, and changes in flow area. Either a pressure rise or a pressure drop may occur from blade root to tip depending on which of these effects predominates. The rise in pressure and temperature of the cooling air from rotor hub to blade root is then used to evaluate the external supply pressure required to pump the cooling air and the supply-air temperature permissible at the turbine-rotor hub to maintain the specified cooling-air temperature at the blade root.

Cooled-turbine operating conditions and performance. - Up to this point in the analysis, the known operating conditions of the uncooled engine are used as a first approximation to determine coolant-flow and coolant-supply conditions. New turbine operating conditions are then computed for the cooled engine and the effect on engine thrust and specific fuel consumption is thereby estimated. The cooling-air-supply pressure that has been determined fixes the bleed point on the engine compressor for the rotor coolant. The stator coolant is bled from the compressor discharge. The power required of the turbine expansion, including the external pumping through the main compressor and the internal pumping through the shroud and blade passage, can then be determined. This power must be extracted from the reduced weight flow of working fluid available to the turbine; therefore the specific turbine work is increased. This increased specific turbine work and the heat transferred from the working fluid to the coolant are used in determining the new turbine-discharge conditions. The mixing of the rotor coolant with the main working fluid further reduces the temperature downstream of the turbine. The net effect of cooling is a reduction in the jet-nozzle total temperature and pressure ratio. The effect of bleeding the compressor to obtain air for cabin pressurization and conditioning, auxiliaries, or deicing is analyzed in reference 4. The method of reference 4, however, is not applicable to this analysis because it does not consider reintroduction of the bleed air into the jet nozzle.

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Assumptions

Many assumptions are necessary in order to make this analysis, although it is believed that the significant variables are considered. The assumptions may be divided into two classes, those used in the development of the analytical methods, and those used in this particular application of the methods.

Assumptions in development of analysis. - The most significant assumptions made in the development of the methods are as follows:

A one-dimensional blade-temperature distribution is used in which blade temperature at any radial cross section is constant. Experiments have shown that the temperatures of the leading and trailing edges of cooled blades are considerably higher than the temperature of the main portion of cooled blades similar to the blades of figure 2, which do not have special provision for leading- and trailing-edge cooling. The leading- and trailing-edge temperature gradients are not considered in determining the required coolant flow, and the blade temperatures obtained are therefore average temperatures over the midchord section of the blade. The analysis makes use of a simplified form of the one-dimensional equation, the derivation of which requires the additional assumptions that the gas-temperature profile is uniform, that the blade-geometry and inside and outside heat-transfer coefficients have a mean value over the span of the blade, and that the radiation and conduction effects can be neglected.

As shown in reference 3, the one-dimensional analysis can be used to consider the effects of the radial variation of inside and outside heat-transfer coefficients, blade geometry, and radial turbine-inlet gas-temperature profile, which is the strongest influence in the blade-temperature distribution. The solution with radial variations, however, requires the use of tedious numerical-integration procedures and for the comparative results desired in this analysis the additional accuracy that could be obtained was unjustified.

As shown in reference 5, neglect of radial conduction to the blade root generally introduces a negligible error in computing

blade-temperature distribution and coolant-flow requirements for blade dimensions such as used in these calculations. Neglect of radiation to the blade introduces some deviation in blade temperatures but has little effect on comparative results. The analyses in reference 5 indicate that at a gas temperature of 1900° F, neglect of the predominant nozzle radiation may result in a uniform
1331 error of 25° F in the spanwise blade-temperature profile when the turbine nozzles are uncooled. (See fig. 8, reference 5.)

The effect of assuming constant geometry, heat-transfer coefficients, and gas properties is illustrated in figure 3 for the finned-blade passage shown in figure 2. A typical radial temperature profile of the gas at the turbine inlet and its effect on the one-dimensional radial blade-temperature distribution computed for different assumptions is presented in figure 3. These results are taken from reference 3. The dot-dash curve represents the case in which radial variation of internal and external geometry, relative gas velocity, internal and external heat-transfer coefficients, and physical gas properties are considered. Assumption of constant mean values for all variables except gas temperature results in the blade-temperature profile given by the dashed line. The computed temperatures are considerably higher near the tip of the blade but are comparable for radial blade stations between 0.2 and 0.6. Both temperature distributions are for a rotor coolant-flow ratio of 0.02, and in both cases the blade-metal temperature distribution resembles the gas-temperature profile. The effects of changes in shape in gas-temperature profile are discussed later inasmuch as it is a design condition that may be imposed.

The physical significance of the assumption of one-dimensional blade-temperature distribution is that the blade configuration would have to be modified to allow, insofar as is possible, uniform cooling of the profile, at the same time retaining the essential performance and flow characteristics of the original uncooled turbines.

Analysis of the effects of heat transfer, friction, centrifugal compression, and geometry in the blade passage is based on one-dimensional equations given in reference 3. It is believed that with the small passages that occur in cooled blades essentially one-dimensional flow will exist. Reference 3 also shows that solution of the equations for Mach number and pressure distribution is practically unaffected by the use of constant heat-transfer coefficients and uniform turbine-inlet gas-temperature profile, as was assumed in this analysis. In determining the passage-flow characteristics, the static pressure that prevails inside the passage at the blade-tip section is assumed equal to the turbine-discharge total pressure and the entire velocity head of the cooling air as it emerges from the tip is assumed lost. In reference 6, it is shown that a large part of the kinetic energy of the cooling air can be recovered as reaction if the blade tips are especially designed for recovery.

In the analysis of cooling losses and their effect on engine operation, several additional assumptions are made. The stator and rotor cooling air are not considered as part of the turbine working fluid but are considered as part of the jet-nozzle gas weight flow. It is also assumed that the compressor operating point is unaffected by the interstage bleed for cooling air and that the stator and rotor blading for the cooled engine can be redesigned in order to maintain matched compressor operation and to extract the higher specific work from the reduced weight flow of working fluid available. 1331

Assumptions in application of analytical methods. - In the application of the analytical methods previously outlined for determining the coolant-flow requirements, additional assumptions were made that have no bearing on the methods or equations but affect the numerical results. It is currently necessary to use blade-profile heat-transfer correlations that have been obtained with static cascades of blades and coolant-passage heat-transfer correlations that apply for tubes. In both cases the effect of centrifugal forces on the heat-transfer coefficient is neglected, and for this reason it is expected that the numerical results may be somewhat optimistic. Regardless of absolute magnitude, however, the use of such correlations introduces Reynolds number effects into the cycle calculations and makes the computed engine variables sensitive to the main parameters that affect heat transfer. It is believed that the trends that are indicated in the results will not be seriously affected when more applicable heat-transfer data become available.

Except where otherwise noted, a flight Mach number of 0.788, a uniform turbine-inlet temperature of 1500° F, an engine speed of 7600 rpm, and a relative total inlet cooling-air temperature of 300° F are used throughout the analysis. An allowance was made for heating of the cooling air as it passed radially outward along the face of the turbine wheel and the required amount of intercooling between compressor and turbine to control the cooling-air temperature was computed. No extra losses due to pressure drop in the intercooler were considered and it was assumed that the heat was rejected to the fuel; thus no calculation of momentum losses due to supplying ram air for the heat exchange were necessary in the analysis of over-all engine performance.

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Design Considerations

The control of blade-temperature distribution is dependent on a number of other factors, which influence the initial design of the engine and the blading. Some of these factors are the geometric configuration of the coolant passage, the radial temperature profile of the gas, and the existence of local temperature gradients in the profile. The manner in which these factors enter the setup and the evaluation of the analysis is indicated in the following section:

Significance of blade-cooling-passage configuration. - The coolant-passage configuration influences the coolant-flow requirements and the pressure drop or pressure rise of the cooling-air flow as it passes through the blade. Ram air is generally not at sufficient pressure to supply the required cooling air flow; the engine compressor must therefore be bled, which in turn complicates the design and affects engine operation.

Three types of air-cooled blade in order of increasing effectiveness are: the plain hollow blade, the hollow blade with insert, and the hollow blade with fins in the coolant passage. The effectiveness is defined as the difference between effective gas temperature and blade-metal temperature divided by the difference between effective gas temperature and inlet coolant temperature at the blade root. Thus the effectiveness of a cooled blade is a measure of the metal-temperature reduction achieved by the cooled blade for a given coolant flow, relative to the temperature reduction ideally possible for given gas and coolant temperatures. The effectiveness is limited by the magnitude of the heat-transfer coefficients and the length of the conductive path through the metal. It can be seen that the effectiveness ratio for a given blade-cooling-passage configuration also expresses the amount by which the effective gas temperature can exceed the metal temperature for a given coolant flow and coolant temperature. Analyses have shown that the effectiveness of the plain hollow blade is very poor, which limits the design to more effective configurations such as the insert and finned blades illustrated in figure 2.

In the insert blade, the effectiveness is augmented by restricting the coolant to the annular passage immediately adjacent to the inside surface of the blade. The improved effectiveness of the finned blade is largely achieved by the increased surface area exposed to the coolant and the conduction of heat to these areas by means of the fins. When the finned blade and the insert blade are compared, in a given case essentially the same amount of heat-transfer and metal-temperature reduction can be accomplished but with considerably less weight flow of cooling air with the finned blade. In the finned blade, however, the air-temperature rise is more rapid and, consequently, the momentum pressure loss of the cooling air is greater. In most cases the friction losses are also higher in a finned blade, and thus the higher effectiveness of the finned blade is offset to some extent by the greater coolant supply pressure that must be maintained. The net effect can be evaluated only by the complete engine analysis, which reflects the effects of compressor bleed and pumping power. A further consideration is the need to compromise between the complexity of the blades having higher effectiveness and their producibility. In many cases, however, a limit exists on the amount of losses that can be sustained, and because of losses the amount of coolant flow must be minimized. It is therefore necessary to develop fabrication methods for the more complex blade types so that the maximum effectiveness possible can be attained.

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Effect of gas-temperature profile. - The effect of radial turbine-inlet-gas-temperature profile on blade-metal-temperature distribution is shown in figure 4. The typical gas-temperature profile is taken from figure 3. The uniform gas-temperature profile shown in figure 4 is an integrated mean of the typical gas-temperature profile. An allowable blade-metal-temperature distribution based on stress-rupture data for a Cr-Mo-Va steel blade is also given.

The blade-temperature distributions for the typical gas-temperature profile and the uniform gas-temperature profile are matched to the allowable curve so that the blade-metal temperature does not exceed the allowable blade-metal temperature at any point. The coolant-flow ratio required for the uniform gas-temperature profile is 0.0145, whereas that required for the typical profile is 0.020. The uniform profile appears to be more desirable because the coolant flow required is lower and the blade is not overcooled, over a large part of the span, to the extent that occurs when the gas-temperature profile varies widely.

Effect of temperature gradients in blade profile. - The possible magnitude of temperature gradients in the trailing edge of the finned-blade profile is shown in figure 5. The spanwise distribution of trailing-edge temperature is compared with the gas-temperature profile and midchord-section-temperature distribution from figure 3. For the typical gas-temperature profile, the trailing-edge temperature reaches 1560° F as compared with a maximum temperature of 1230° F at other parts of the profile where cooling is more directly applied. The high trailing-edge temperature is a result of the long conduction path from the trailing edge to the cooling-air passage. The desirability of blades with short trailing edges is apparent. Careful design is required to minimize hot spots; and in the compromise between blade-profile aerodynamics and heat transfer, reduction in required coolant-flow and trailing-edge temperature by means of a blade shape that favors cooling is believed to justify some loss in turbine efficiency that will probably result. The most aerodynamically efficient turbine does not necessarily result in the highest over-all engine efficiency.

Determination of Allowable Blade-Temperature Distribution

Another important variable, the determination of which depends on a number of assumptions, is the allowable blade-metal-temperature distribution for a desired blade life. Typical factors that should enter this determination are the combined stresses due to centrifugal forces, gas bending forces, vibratory excitation, and the strength properties of the blade materials under these combined stresses. The influence of thermal cycles and corrosion effects should also be correlated, and all these variables should be systematically related to the alloy content of the blade materials, as well as to the life of the blade. No satisfactory method has as yet been devised whereby all these factors can be taken into account.

Specification of allowable temperature. - It is currently necessary to relate blade life to operating stress and temperature by means of the stress-rupture characteristics of the materials, which vary widely with the alloy content. The simplest centrifugal stress is used as the criterion in determining an allowable temperature distribution. The assumptions used to determine the allowable blade-metal temperature have no effect on the methods of analysis used to determine the required coolant flow for any specified allowable temperature. The effect of a given coolant flow on engine performance is completely independent of the factors that enter the determination of an allowable temperature. The use of rupture criterion specifies the upper limit of blade life and the lower limit of required coolant flow for any selected condition. Actual blade life will probably be much shorter because of other recognized influences such as fatigue, thermal effects, and corrosion phenomena, which at present must be evaluated from experience.

The rupture properties of representative alloys, which are used to define the allowable blade temperature, are illustrated in figure 6, which presents allowable stress against temperature for a rupture life of 1000 hours and the cooling requirements for a representative engine condition. Of the alloys shown, the highest properties are developed by S-816, a high-temperature alloy that contains approximately 96-percent strategic metals and about 4-percent iron, which is suitable for uncooled operation at 1500° F at the stress level of about 16,500 pounds per square inch shown by the dot-dash line. This stress level is arbitrarily set in figure 6 for the purpose of comparing other materials with S-816 operating uncooled at 1500° F on the basis of 1000-hour rupture life. At the low end of the alloy scale is SAE 1015, which is substantially 100-percent iron; for this stress level, SAE 1015 has an allowable temperature of about 970° F, which is more than 500° lower than the uncooled blade. As a first step in reduction of strategic-metal content, an intermediate group of alloys containing 50 percent or more iron and eliminating the most strategic metals, columbium and cobalt, may be considered. Typical of these alloys is the 16-25-6 alloy, which, for the specified stress level of 16,500 pounds per square inch, has an allowable temperature of about 1380° F and requires less than 125° F reduction in temperature below the uncooled blade of S-816.

In the range between SAE 1015 and the 16-25-6 alloy, there are many ferrous alloys that develop remarkable properties with very small additions of strategic alloying elements and thus offer possibilities of substantial savings in strategic elements. A typical alloy in this range is Cr-Mo-Va steel having less than 5-percent alloy content and an allowable temperature around 1100° F at the stress level of 16,500 pounds per square inch. The properties of this steel are superior to many highly alloyed stainless steels at temperatures up to 1100° F. New developments such as the one point shown in figure 6 for the Ti-Bo (titanium-boron) steel indicate that further improvements may be expected. This particular steel has 2.25-percent chromium and 1-percent molybdenum, with additions of 0.30-percent titanium and 0.03-percent boron.

Limitations in allowable blade temperature. - The significance of diminishing returns in the cooling process is apparent from the lower curve in figure 6, which illustrates the rotor coolant-flow ratios required for the arbitrary stress level starting with 1500° F as the uncooled-blade temperature. The coolant-flow ratios are plotted against blade-metal temperature at the midspan of a seven-finned blade assuming a constant value of thermal conductivity for all metals shown. The vertical dotted lines indicate the allowable temperatures for 1000-hour rupture life for the various blade materials at a stress of 16,500 pounds per square inch. The coolant-flow requirement increases rapidly with temperature reduction below that of the uncooled blade. For example, the 100° F decrement from 1500° to 1400° F is obtained with an increment of approximately 0.0025 in coolant-flow ratio; whereas the 100° F decrement from 1000° to 900° F requires a corresponding increment of 0.0135 in coolant-flow ratio. The coolant flow required to achieve the 400° F reduction in metal temperature for the Cr-Mo-Va steel is a little more than one-half that required for the SAE 1015 steel. Thus, eliminating the remaining small amount of alloying elements results in approximately twice the coolant-flow requirements for a fixed turbine-inlet temperature of 1500° F, and it is evident that a compromise is necessary in establishing a permissible alloy content for the nonstrategic-alloy blade.

The low coolant-flow ratios that are illustrated in the lower curve apply to an actual engine only for design conditions that fulfill those of the calculation. These low coolant-flow ratios could be obtained in practice only for ideal conditions of gas-temperature distribution and blade stresses and for a modified blade profile and passage configuration that provide a favorable blade-temperature distribution approaching the one-dimensional case. In this sense, each point on the curve represents a design point and is the minimum coolant-flow ratio that could occur under the specified operating conditions of the turbine. Similarly, the design conditions that result in these minimum coolant flows represent a goal or objective that should be considered in the initial design of a cooled turbine.

In the results subsequently given, various curves are identified by the alloy that was used to specify the allowable metal temperature; it is necessary to recall that practical realization of a turbine design using the indicated coolant-flow ratios and metals is subject to the limitations discussed throughout the section ANALYSIS.

RESULTS AND DISCUSSION

The methods and the considerations previously outlined have been applied to compare the effectiveness of a finned and an insert blade, similar to those shown in figure 2, using the substitution of nonstrategic blade alloys as a means of selecting the allowable blade-temperature distribution. The typical results given up to this point for the finned blade show effects that occur at some constant specified flight condition. Over a range of flight conditions, altitude and flight Mach number effects are encountered because the blade heat-transfer coefficients are dependent on the variations in engine-gas weight flow and pressure ratio that occur with altitude and flight speed.

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Coolant-Flow Requirements

The coolant-flow ratios required for the finned and insert blade of nonstrategic Cr-Mo-Va steel are presented in figure 7(a). The coolant-flow ratio for both finned and insert blades increases with altitude, for example from 0.0105 at sea level to 0.018 at the 40,000-foot altitude for the finned blade. The lower effectiveness of the insert blade is shown by the substantially larger coolant-flow ratios, almost twice that of the finned blade, throughout the altitude range. Finned blades with smaller fin thickness and spacing than the blade used in this analysis have been made for experimental purposes and have greater theoretical cooling effectiveness than the blade considered in this report. The Mach number through the coolant passage also increases with coolant-flow ratio and altitude and both of these factors tend to increase the coolant-supply pressure required for the cooling air. A critical altitude might be defined for a given blade where the required coolant supply pressure reaches the compressor-discharge pressure, although this limitation could be relieved with an auxiliary compressor. Blades with high effectiveness and with convergent cooling passages tend to have a higher pressure drop than blades of low heat-transfer characteristics and constant-area passages. It is therefore unwise to establish the mechanical and aerodynamic design of the blading without considering passage configuration inasmuch as pressure limitations may be encountered later when attempting to provide adequate cooling. No detailed analysis has been made of the potentialities of air cooling at altitudes above 40,000 feet. In general a cooling-air pressure ratio equal to the turbine pressure ratio is always available by bleed from the compressor discharge.

Engine Performance

A comparison of blade configurations on the basis of required coolant-flow ratio alone does not include the adverse effects of the higher pressure drop in the finned blade. In order to make this comparison, it is necessary to complete the engine calculations to determine the thrust and the specific fuel consumption. The comparison of the two configurations on the basis of percentage loss in specific engine thrust at the required coolant-flow ratio is presented in figure 7(b) for the same flight conditions and for the fixed turbine-inlet temperature of 1500° F. The finned blade has the lower thrust loss throughout the altitude range, and at a maximum altitude of 40,000 feet shows a thrust loss of approximately 3.5 percent as compared with 5.5 percent for the insert blade. The percentage loss in specific thrust does not completely reflect the change in over-all engine efficiency brought about by the cooling losses inasmuch as the air bled from the compressor for cooling is not burned and thus less fuel is used in the cooled engine. It is, however, an important performance variable that is considered in evaluating cooling.

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A truer measure of the cooling losses is the percentage increase in specific fuel consumption, which bears a direct relation to the over-all efficiency of the engine. The percentage increase in fuel consumption is plotted against altitude for the two cooling-passage configurations in figure 7(c). The specific fuel consumption is increased significantly with both blades and the net losses with the finned blade are less than with the insert blade.

The manner in which the cooling variables influence the thrust and fuel consumption is shown in table I, which presents the significant items in the comparison among the finned blade, the insert blade, and the uncooled engine.

Although the coolant-passage pressure ratio, which represents the pressure drop of the cooling air, is larger for the finned blade than for the insert blade, the external coolant compressor horsepower is less for the finned blade. This condition results from the smaller weight flow of coolant required. Likewise, the internal coolant pumping power for the finned blade is lower than for the insert blade. In addition, the weight flow of working fluid available in the turbine is higher with the finned blade. The work required per pound of turbine working fluid consequently is less with the finned blade, which results in higher jet-nozzle total-temperature and pressure ratios.

Similar comparisons have been made over a wide range of conditions, and at higher coolant flows more marked differences occur between the finned and insert blades. A comparison with the results of table I is given in the following table for higher required coolant flows, which represent a reduction of approximately 100° F in the allowable blade-metal temperature relative to that of table I. The other engine operating conditions are the same as those of table I.

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	Required coolant flow of table I		Higher required coolant flow (lower allowable blade temperature)	
	Finned blade	Insert blade	Finned blade	Insert blade
Rotor coolant-flow ratio	0.018	0.034	0.030	0.063
Decrease in thrust, percent	3.5	5.5	6.3	11.0
Increase in specific fuel consumption, percent	0.70	1.15	2.47	4.07

For either the finned or insert blade, the increase in required coolant-flow ratio results in a substantial increment in specific fuel consumption. At the higher coolant-flow ratios, the advantage of the finned blade is indicated by the much larger differences in performance.

As previously mentioned, these results are the minimum coolant-flow ratio and cooling losses that could occur and other factors such as gas-temperature profile, large local temperature gradients, and the combined stresses, which occur in actual engine operation, would result in substantial increases in coolant flow and cooling losses. This fact reemphasizes the necessity for attempting to approach the design conditions of gas-temperature profile and favorable blade-temperature distribution previously mentioned.

Heat-Exchanger Requirements

The desirability of a heat exchanger between the compressor bleed point and the turbine to control the cooling-air temperature was previously mentioned. The emphasis has previously been placed on coolant-flow ratio; however, the cooling-air temperature is also of importance. In addition to the saving in cooling-air-flow requirements permitted by lower inlet cooling-air temperatures, the required coolant-supply pressure is also reduced due to the decreased cooling-air Mach number in the passage.

The required coolant-flow ratios given herein are all for an assumed relative total cooling-air temperature of 300° F in the cooling passage at the blade root. The compressor-air temperature at the bleed point is, in most cases, higher than that allowed by the specified cooling-air temperature of 300° F at the blade root and a heat exchanger would be required. For the two configurations considered herein, the maximum heat dissipation is required for the insert blade at sea level and is approximately 63,000 Btu per hour. This dissipation is less than the probable oil-cooling requirement 1331 of the engine so that the problem of heat disposal is not critical in supplying cooling air to the blade root at 300° F. Further reduction in cooling-air temperature would be desirable for the reasons previously given.

Effect of Increased Turbine-Inlet Temperature on Required Coolant-Flow Ratio and Engine Performance

As turbine-inlet temperature is increased in an uncooled turbojet engine, both thrust and specific fuel consumption increase. The results previously given have shown that air cooling of the turbine blades adversely affects the engine performance. The calculations were therefore extended for the 40,000-foot-altitude flight condition to determine coolant-flow requirements and engine performance at turbine-inlet temperatures of 2000° and 2500° F. The general engine configuration, engine speed, compressor-air weight flow, and compressor pressure ratio were held constant. It was also assumed that the turbine blading could be modified to match the compressor for all coolant-flow ratios and turbine-inlet temperatures. In evaluating the engine performance at the 2000° and 2500° F turbine-inlet temperature, it was assumed that the cooling air was bled off at the compressor discharge. Otherwise, the same assumptions that were made in the analysis for a turbine-inlet temperature of 1500° F were made for the high-temperature calculations. At higher turbine-inlet temperatures, the neglect of radiation effects is not as reasonable an assumption as at the gas temperature of 1500° F and would make the calculated coolant flows lower than required if radiation were considered; however, this assumption has no effect on the performance values obtained for a given coolant flow.

The values of thrust and specific fuel consumption obtained in this analysis are plotted in figure 8. The thrust is plotted in figure 8(a) as percentage of that of the uncooled engine at a turbine-inlet temperature of 1500° F against total coolant-flow ratio for three turbine-inlet temperatures. For each turbine-inlet temperature, the thrust decreases with increasing coolant-flow ratio. Turbine-inlet temperature can be increased to overcome this loss; but as a result of this gas-temperature increase, the coolant-flow requirement for a given blade metal also increases causing additional cooling losses. This fact is illustrated by the two curves for finned blades of Cr-Mo-Va steel and SAE 1015. For the Cr-Mo-Va steel blade at a turbine-inlet temperature of 1500° F, the total, or rotor-plus-stator, coolant-flow ratio requirement is 0.028 and the thrust decreases to 96.5 percent of rated thrust. To restore the engine to rated thrust requires an increase in turbine-inlet temperature that results in an increase in required coolant-flow ratio to approximately 0.034, as indicated by the point of intersection of the Cr-Mo-Va steel curve and the curve of 100-percent rated thrust. For the blade of SAE 1015, a larger increase in turbine-inlet temperature is necessary to restore the thrust and is accompanied by a still larger increase in required coolant-flow ratio.

In completely eliminating the strategic-alloy content in the blades, for example, by substituting SAE 1015 for a low-alloy steel, alloy requirements of other engine parts that are also exposed to the hot gases and represent a larger percentage of the total engine weight may be increased as a result of the higher turbine-inlet temperature. In an engine using nonstrategic materials, the rotor blades should have the highest alloy content of any of the parts exposed to the hot gases in order to achieve maximum saving of strategic metals.

At a turbine-inlet temperature of 2000° F, an engine with Cr-Mo-Va steel blades requires a total coolant-flow ratio of 0.075 and has a net thrust 23 percent greater than the uncooled engine at 1500° F. At a turbine-inlet temperature of 2500° F with the Cr-Mo-Va steel blade, a total coolant-flow ratio of 0.130 is required and a net thrust 43 percent greater than that of an uncooled engine at 1500° F is obtained.

A plot of specific fuel consumption against total coolant-flow ratio for three gas temperatures is shown in figure 8(b). For an uncooled engine, the specific fuel consumption increases as turbine-inlet-gas temperature is increased. Also, for all three gas temperatures, the trend of specific fuel consumption is to increase as coolant-flow ratio is increased. At low gas temperatures, the specific fuel consumption is sensitive to low coolant-flow ratios; whereas at higher gas temperatures larger coolant-flow ratios can be used without excessive increases in specific fuel consumption relative to a coolant-flow ratio of 0. This relation probably exists because for a given coolant flow the cooling losses are a smaller percentage of the higher total energy available at higher temperatures.

The following table gives some results of the calculations made of engine performance at a turbine-inlet temperature of 2500° F with uncooled turbine blades and with finned and insert blades. The coolant flows used with the finned and insert blades are the calculated required coolant flows for Cr-Mo-Va steel blades with 1000-hour rupture life. The calculations were made for a flight Mach number
1331 of 0.788, an altitude of 40,000 feet, and an engine speed of 7600 rpm.

	Type of blade		
	Uncooled	Cooled	
		Finned	Insert
Required rotor coolant-flow ratio	0	0.120	0.235
Specific turbine work, Btu/lb	72.9	90.5	111.8
Turbine pressure ratio	1.61	1.83	2.14
Jet-nozzle pressure ratio	3.72	3.27	2.80
Jet-nozzle total temperature, °F	2267	2029	1802
Thrust, lb	1575	1359	1137
Specific fuel consumption, lb/lb-hr	1.648	1.662	1.724

Comparison with table I shows that the specific turbine work is higher for the cooled engine at a turbine-inlet temperature of 2500° F than at a temperature of 1500° F as a consequence of the higher pumping power and reduced working fluid available to the turbine. Because the turbine-inlet temperature is higher, however, the required turbine pressure ratio is lower and the jet-nozzle temperature and pressure ratio are higher. The net result is a thrust increase of 48.2 percent for the finned blade and 26.6 percent for the insert blade. The fuel consumption is substantially increased over that obtained at a turbine-inlet temperature of 1500° F, but comparison of the cooled with the uncooled engine at 2500° F shows that the fuel consumption of the finned blade is only slightly greater than that obtained for an uncooled engine at 2500° F at this compressor pressure ratio. The engine with insert blades shows about 4.6 percent increase in fuel consumption when compared with the uncooled engine at the same turbine-inlet temperature.

In the previous calculations for higher turbine-inlet temperatures, it was assumed that, as turbine-inlet temperatures are increased, the compressor-air weight flow and pressure ratio are held constant and that the turbine blading could be modified to match the compressor and the turbine. For an engine having a fixed turbine-stator configuration, a certain percentage of compressor-air weight flow must be bled off in order to match the turbine to the compressor at turbine-inlet temperatures above the design point. Whenever the required bleed for cooling purposes is less than that required for matching, it is desirable to find means by which this difference in weight flow can be passed through the turbine. There are two possible solutions. The compressor pressure ratio could be allowed to increase as temperature is increased and the resulting recovery in gas density would aid in passing the weight flow through the choked stator. The limiting value of compressor pressure ratio at a given rotative speed would be reached at the point of compressor surge. The other possible solution is to incorporate an adjustable-angle turbine stator, such as proposed in reference 7. This variable-angle turbine stator could be adjusted along with increasing turbine-inlet temperature to maintain matching between compressor and turbine.

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Possibilities of Turbojet Engine

The present justification of the turbojet engine in military aircraft is primarily the high thrust of the engine. In general, further increase in engine size to achieve greater thrust does little to improve engine flexibility. The necessity for establishing the aircraft and engine design at a fixed high-power point results in high cruising fuel consumption because the aircraft is forced to fly at an uneconomical high speed.

The performance potentialities of uncooled turbojet engines at various design points is illustrated in figure 9 (data from reference 1), which presents the specific thrust and specific fuel consumption as functions of compressor pressure ratio and turbine-inlet temperature. The relative ultimate range of a suitable high-speed-aircraft configuration is cross-plotted on the figure.

At any fixed compressor pressure ratio within the range of compressor pressure ratios considered, the longest cruising range at subsonic speeds is obtained with the lowest turbine-inlet temperature. However, the fact that there is little margin of excess power available for take-off, climb, and combat maneuvers with such an engine may make it impractical. Figure 9 also shows that much higher specific thrust can be obtained at high turbine-inlet temperature; but the maximum range of the aircraft, for a given compressor pressure ratio, is less. A worthwhile objective would be to evolve a power plant for modern

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high-speed aircraft that would have the desirable features of both high- and low-temperature operation, thereby increasing both the maximum-power performance and the maximum-range performance.

This objective could be realized with an engine design that permits operation at constant compressor pressure ratio and weight flow regardless of turbine-inlet temperature and that embodies a turbine-cooling system that permits selection of any turbine-inlet temperature within an adequate range. The essential design features are a compressor having high peak pressure ratio, for example 16, and an adjustable blade-angle turbine stator that permits matching of the components at any turbine-inlet temperature within the desired range without difficulties from compressor-surge characteristics. The adjustable turbine stator provides a variable flow area between the compressor and the turbine and in conjunction with a variable-area tail pipe gives the turbojet engine many of the desirable cruise-control characteristics of the reciprocating aircraft engine, depending on the effectiveness of the variable-area stator.

If such an engine were designed for operation at a maximum turbine-inlet temperature of 2540° F and a compressor pressure ratio of 16, the performance at all design points shown by the dashed line in figure 9 for a pressure ratio of 16 would be available by proper adjustment of the variable-angle turbine stator. The maximum specific thrust would be in the order of 70 percent greater than that of current engines with considerable improvement in specific fuel consumption. With the use of the adjustable blade-angle turbine stator, the turbine-inlet temperature could be reduced for cruising at a lower power with approximately 23 percent further reduction in specific fuel consumption. The fuel consumption at the cruising condition would be reduced approximately 33 percent relative to engines using a compressor pressure ratio of 4 and a turbine-inlet temperature of 1540° F. The power regulation available in this case would have the additional advantage of permitting an appropriately designed aircraft to operate at higher lift-drag ratios at reduced flight speeds, contrary to the limitations currently encountered with turbojet-powered aircraft. This mode of operation would provide an additional advantage in maximum range while retaining the large margin in excess power available. This power reduction for cruising could be accomplished with an improvement in specific fuel consumption, which is the opposite of the case for the normal engine matched at only one design point. The advantage of this arrangement for maximum endurance, as in airport-traffic patterns, is also apparent in comparison with the current fixed-stator turbojet engine, which does not yet permit economical "stacking" operation. The foregoing presentation is necessarily brief but serves to indicate the nature of future developments that may lead to substantial improvements in over-all aircraft performance.

The application of the turbojet engine at very high flight speeds presents a different problem. At supersonic speed, any engine variable that affects the physical size of the fuselage or other body enclosing the engine is of prime importance. The two most significant results of range analysis at supersonic speeds are that the compressor pressure ratios required are lower and that for any given compressor pressure ratio above 4 an optimum turbine-inlet temperature exists for greatest range, which is a higher temperature than that currently possible with uncooled turbines. This circumstance results directly from the high thrust per square foot of frontal area that is obtained with high turbine-inlet temperatures. 1331

SUMMARY OF RESULTS

Typical results from an analysis of air-cooling the turbine blades of an axial-flow turbojet engine have been presented to illustrate the application of theory and experimental data and to outline the characteristics of the cooling process, the limitations that may occur, and some of the design considerations. These results may be summarized as follows:

1. For a fixed turbine-inlet temperature, air cooling decreased the specific thrust of the engine. The effects of air cooling on specific fuel consumption were considerably smaller.

2. The occurrence of diminishing returns with increase in degree of cooling was observed as a fundamental characteristic and suggested that complete elimination of the strategic elements from the blade alloy is inadvisable on the basis of cooling requirements and losses.

3. The highest possible cooling effectiveness was desirable to minimize losses in engine performance.

4. The required coolant-flow ratio increased with altitude, and some compression of the cooling air was generally required for the cooling-air supply.

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5. The radial turbine-inlet-temperature distribution seemed to have a considerable effect on coolant-flow requirements, and the desirable distribution seemed to be uniform rather than increasing radially with a peak occurring near the blade tip.

6. Because of the high coolant-supply pressure required, it was generally necessary to bleed the cooling air from the compressor.

7. In selecting the blade profile, considerable attention must be given to the cooling of hot spots such as the leading and trailing edges, which otherwise may run several hundred degrees hotter than the midchord section of the blade.

8. The analyses indicated that substantial reduction of strategic-metal content in turbine-blade alloys can be made with reasonable sacrifice in over-all engine performance.

9. In military engines using nonstrategic materials, the rotor blades should still have the highest alloy content relative to the other parts exposed to the hot gases.

10. The application of cooled turbines with high pressure-ratio compressors and adjustable-blade-angle turbine stators offers improvement in flexibility, maximum range, and endurance.

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TABLE I

COMPARISON OF SIGNIFICANT PERFORMANCE VARIABLES FOR COOLED AND UNCOOLED TURBOJET ENGINE

[Altitude, 40,000 feet; flight Mach number, 0.788; turbine-inlet temperature, 1500° F; Cr-Mo-Va steel, 1000-hour rupture life.]

	Type of blade		
	Uncooled	Cooled	
		Finned	Insert
Compressor inlet weight flow, lb/sec	20.14	20.14	20.14
Required rotor-coolant weight flow, lb/sec	- - -	0.36	0.69
Rotor coolant-flow ratio	- - -	0.018	0.034
Stator coolant-flow ratio	- - -	0.010	0.010
Rotor coolant-temperature rise, °F	- - -	624	379
Heat loss to coolant, hp	- - -	80.6	91.6
Coolant-passage pressure ratio	- - -	1.33	1.20
External coolant compressor power, hp	- - -	46.3	65.9
Internal coolant pumping power, hp	- - -	26.0	49.6
Main compressor power plus auxiliaries, hp	2152	2092	2058
Total turbine power, hp	2152	2164	2173
Turbine weight flow, lb/sec	20.50	19.92	19.60
Specific turbine work, Btu/lb	74.2	76.8	78.4
Turbine pressure ratio	2.15	2.21	2.25
Jet-nozzle pressure ratio	2.79	2.72	2.67
Jet-nozzle total temperature, °F	1236	1215	1199
Thrust, lb	950	917	898
Specific fuel consumption, lb/lb-hr	1.328	1.337	1.343
Decrease in thrust, percent	- - -	3.5	5.5
Increase in specific fuel consumption, percent	- - -	0.70	1.15

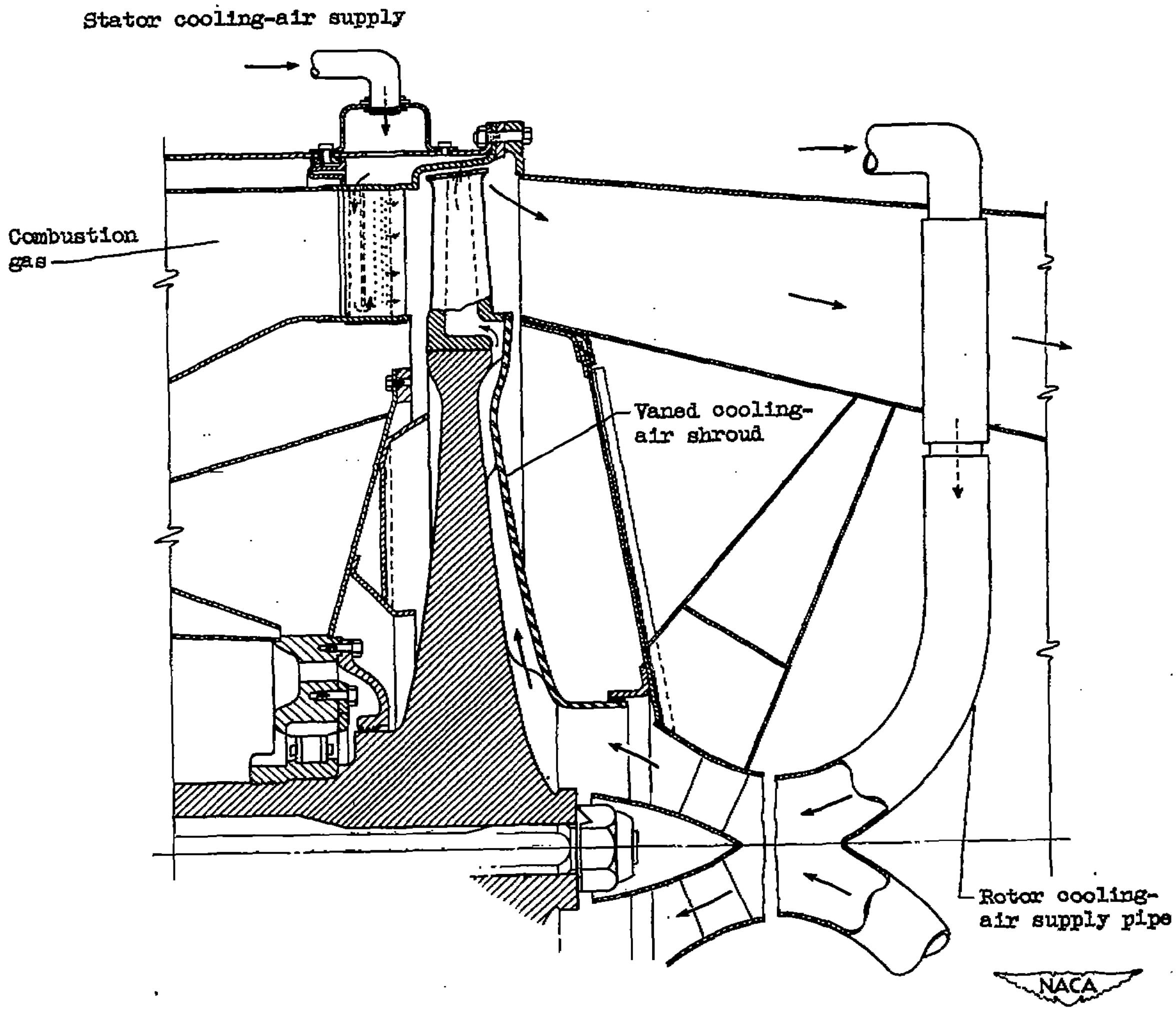

Figure 1.- Section of turbine rotor showing proposed method of introducing cooling air to rotor and stator.

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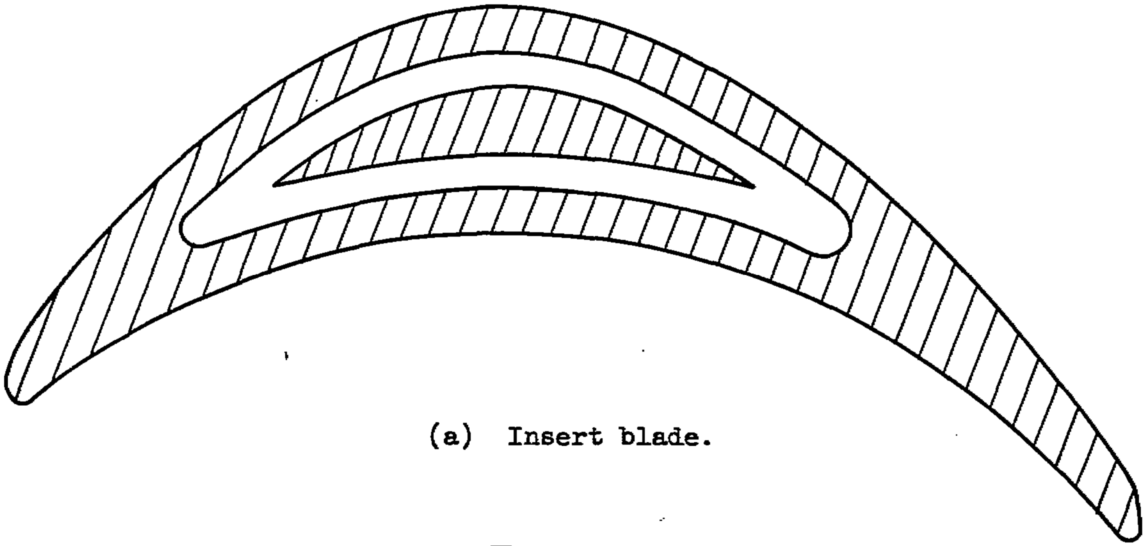
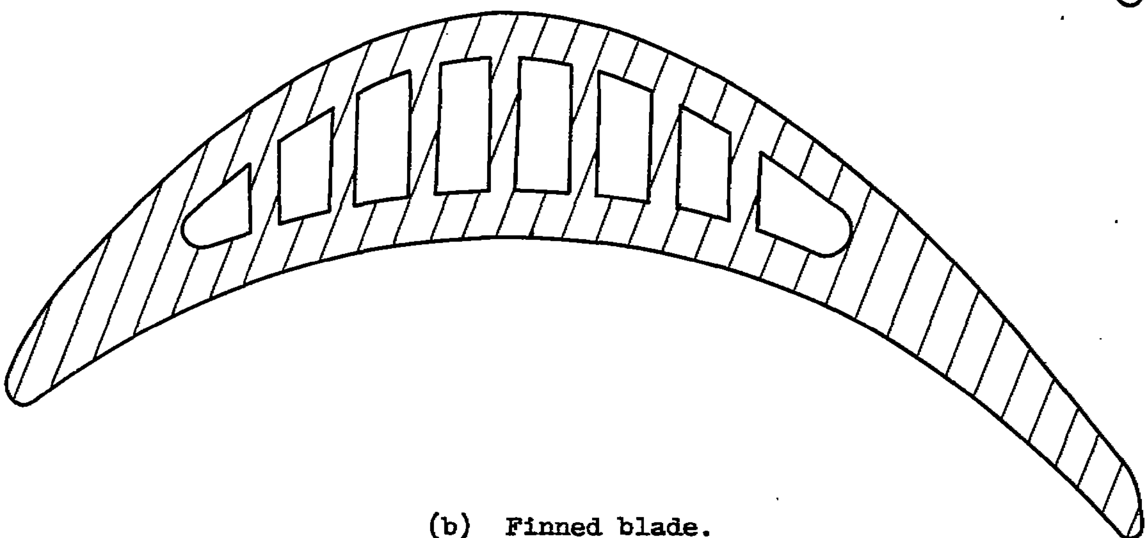


(a) Insert blade.



(b) Finned blade.

Figure 2. - Cross sections of hollow air-cooled blades using two methods of increasing cooling effectiveness.

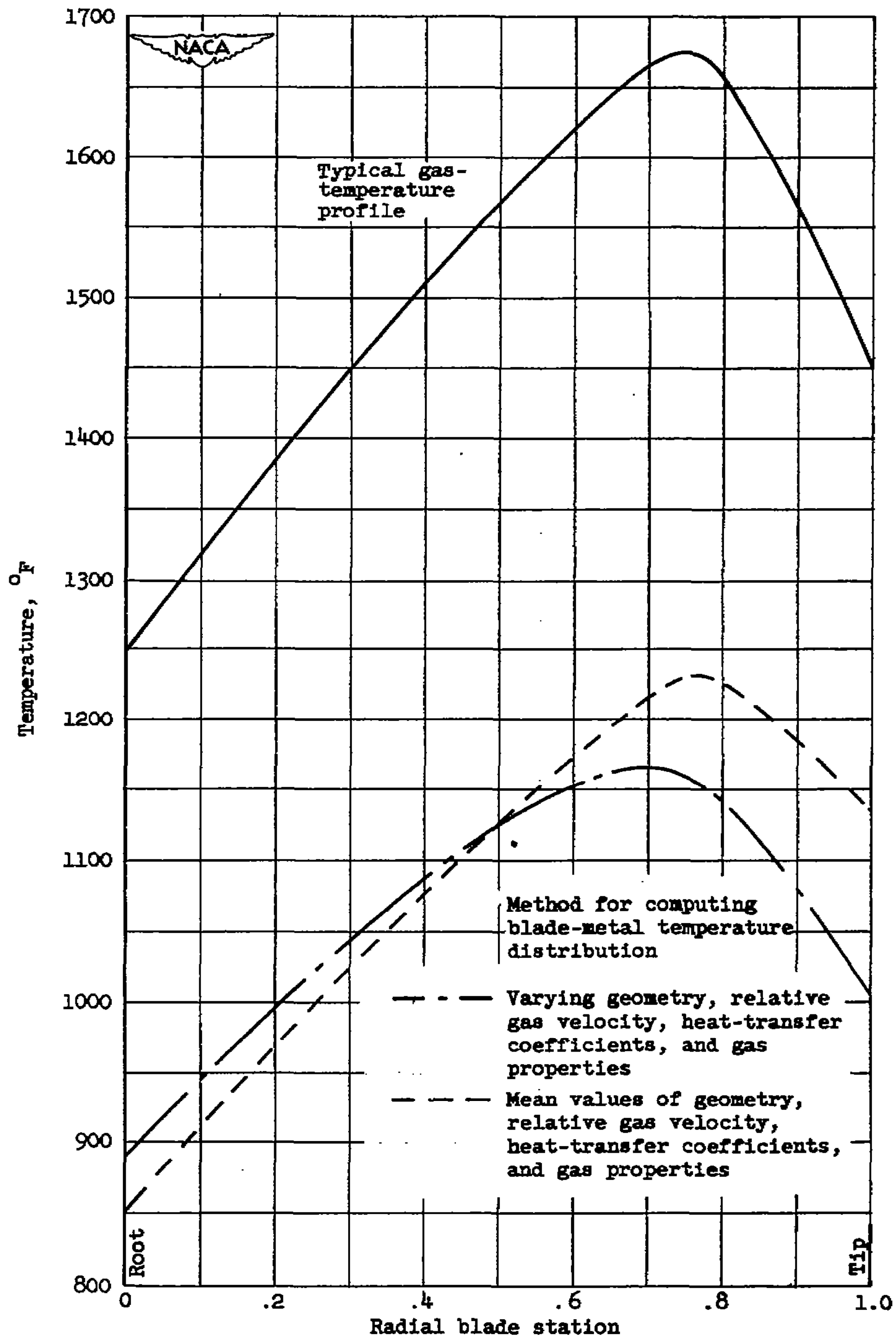


Figure 3. - Comparison of two methods for computing blade-metal temperature distribution using gas-temperature profile shown. Finned blade; altitude, sea level; engine speed, 7600 rpm; rotor-coolant-flow ratio, 0.02. (Data from reference 3.)

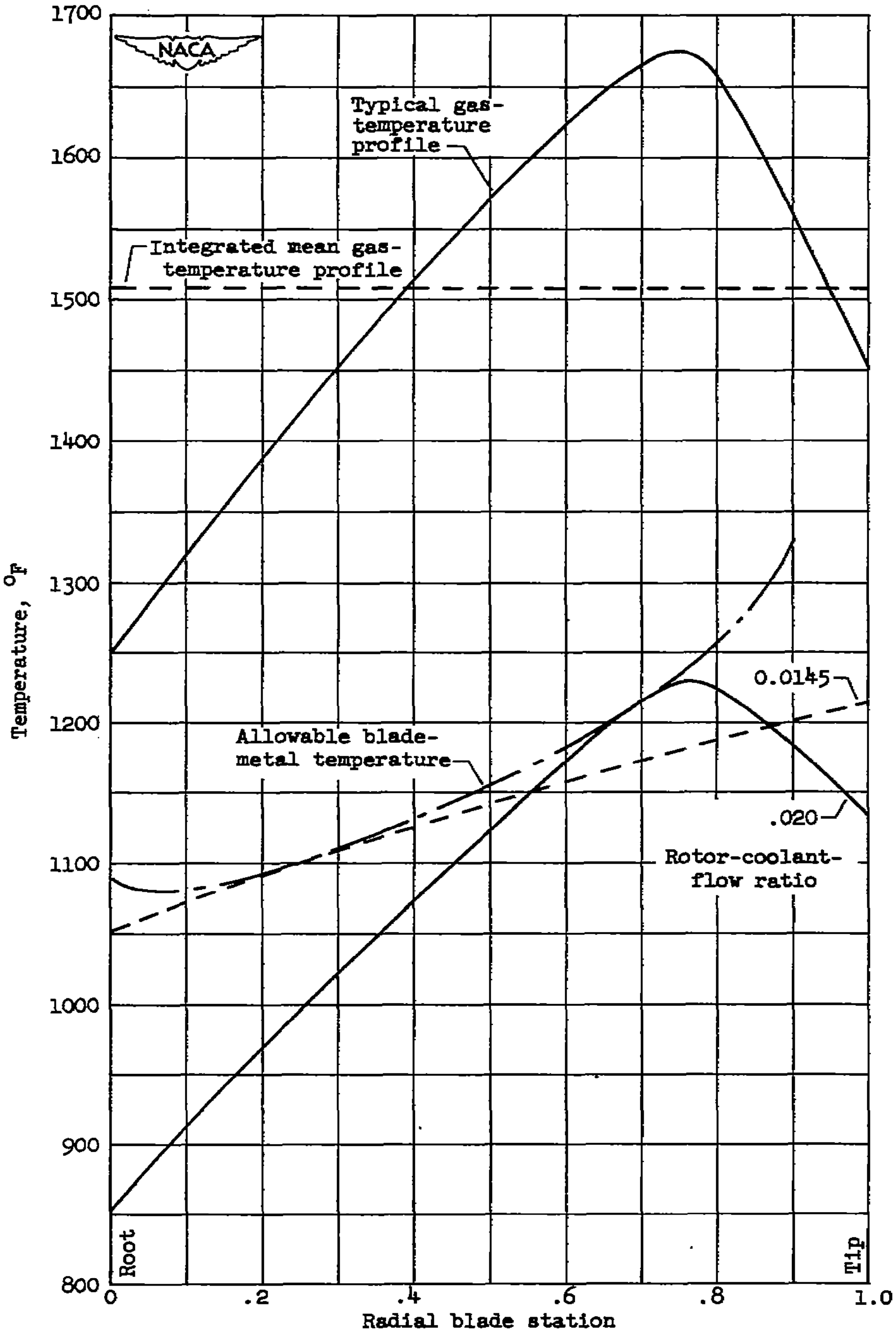


Figure 4. - Comparison of coolant-flow requirement and blade-metal-temperature distribution for two gas-temperature profiles. Finned blade; altitude, sea level; engine speed, 7600 rpm. Allowable blade-metal-temperature curve determined for Cr-Mo-Va steel with rupture life of approximately 4000 hours. (Typical gas-temperature profile and corresponding blade-metal distribution curve from fig. 3.)

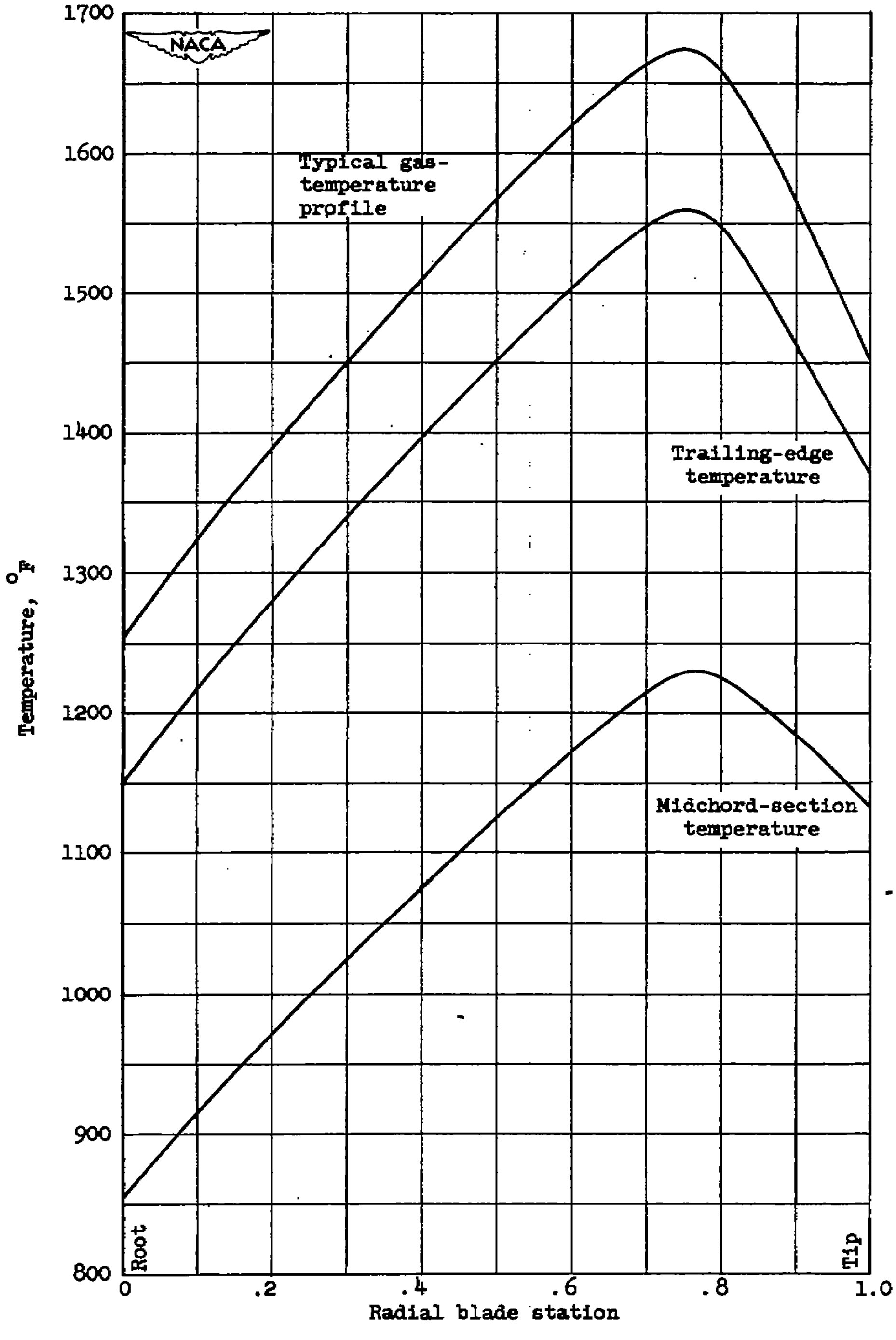


Figure 5. - Approximate trailing-edge-temperature distribution for gas-temperature profile and midchord-section temperature shown. Finned blade; coolant-flow ratio, 0.02; thermal conductivity, 15 (Btu/(hr)(ft)(°F)). (Typical gas-temperature profile and midchord-section-temperature curve from fig. 3.)

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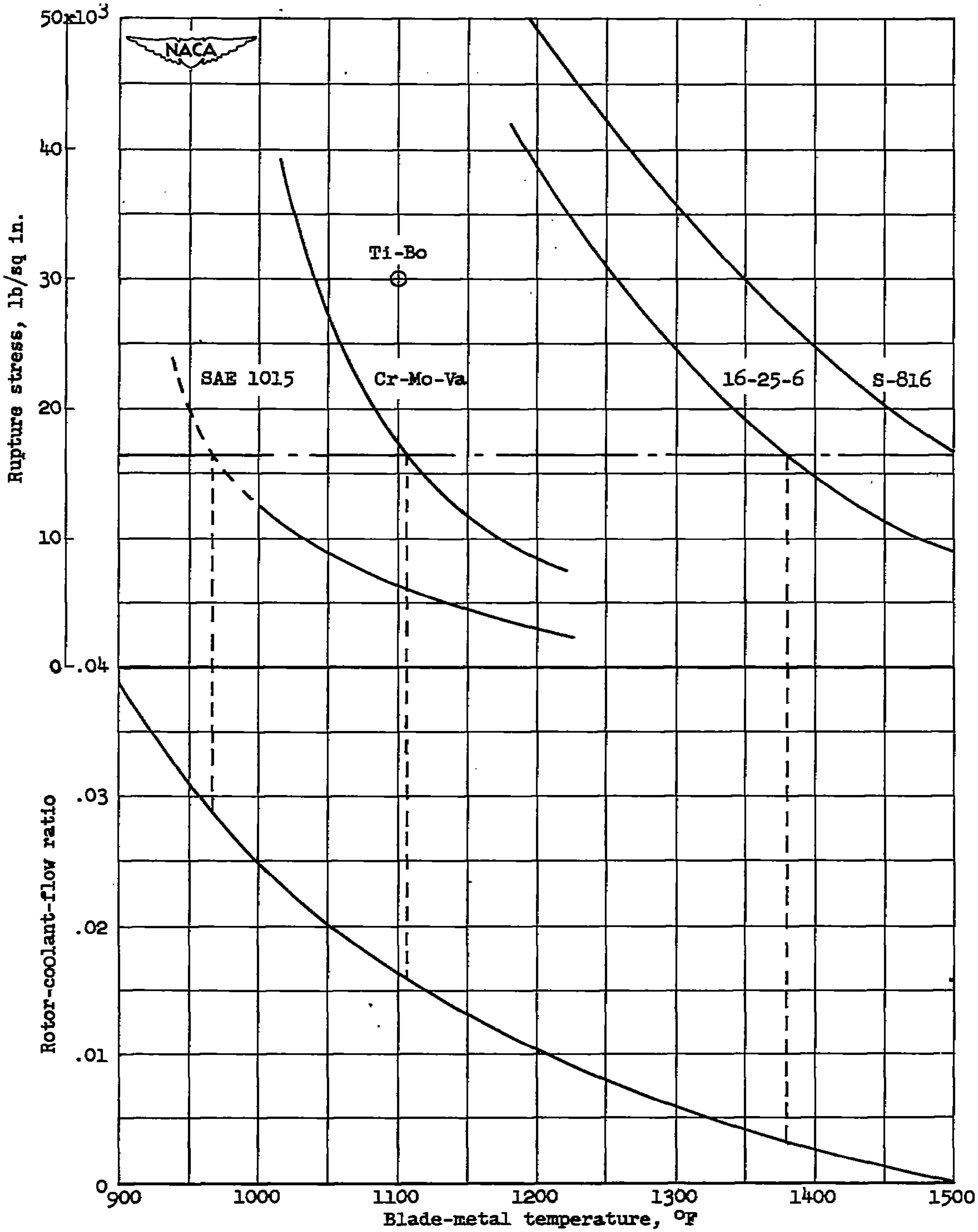
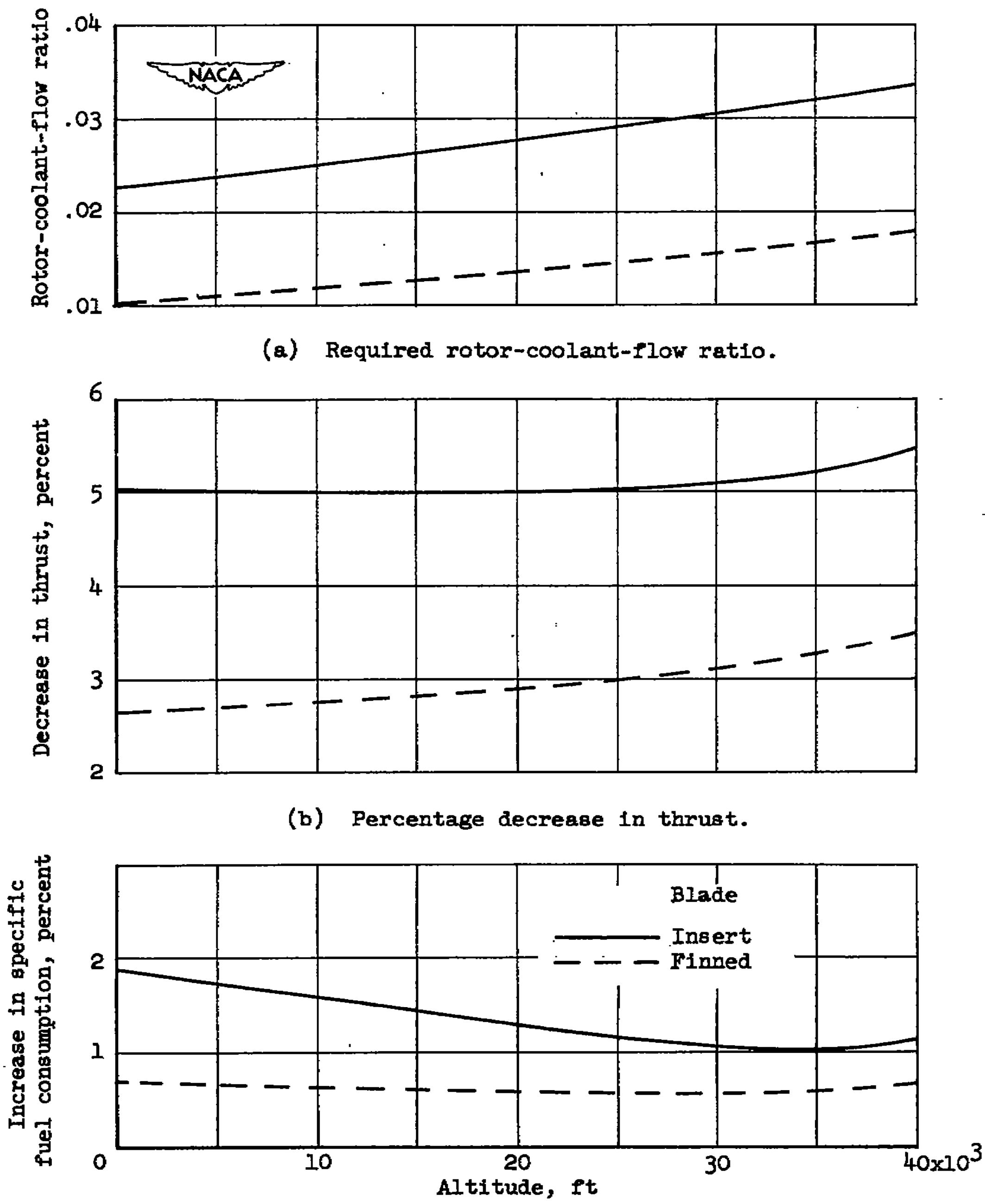


Figure 6. - Comparison of stress-rupture properties of various blade materials for 1000-hour blade-rupture life. Rotor-coolant-flow requirements against blade-metal temperature at midspan for finned blade. Altitude, 40,000 feet; flight Mach number, 0.788.



(c) Percentage increase in specific fuel consumption.

Figure 7. - Required coolant flow, percentage decrease in thrust, and percentage increase in specific fuel consumption for finned and insert blades. Blade material, Cr-Mo-Va steel; blade-rupture life, 1000 hours; flight Mach number, 0.788; turbine-inlet temperature, 1500° F.

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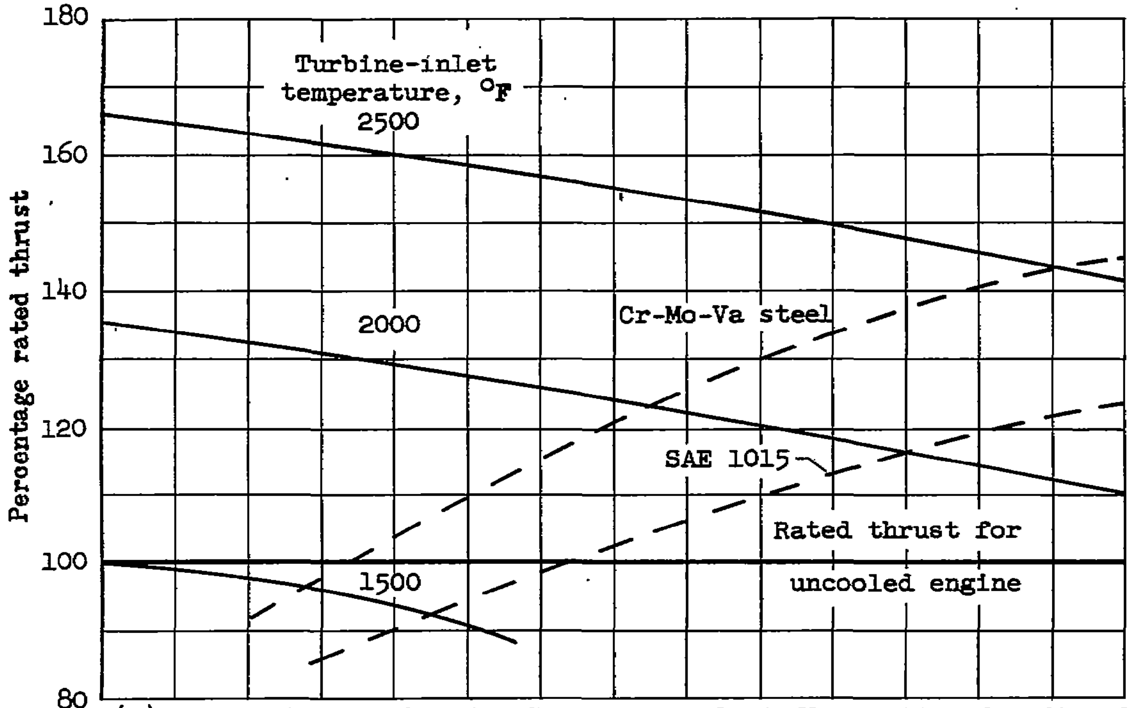
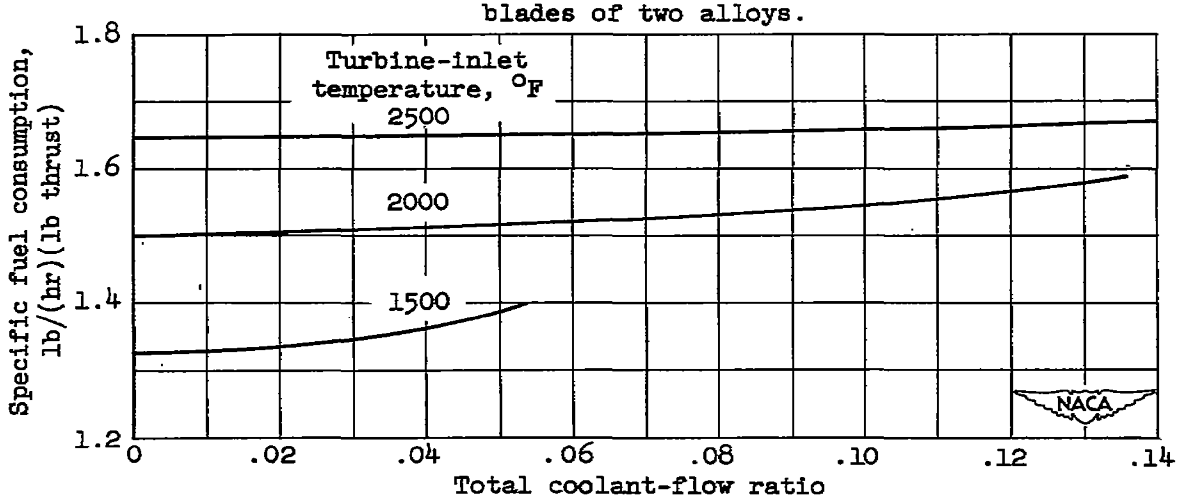


(a) Variation of thrust. Required coolant-flow ratios for finned blades of two alloys.



(b) Variation of specific fuel consumption.

Figure 8. - Variation of thrust and specific fuel consumption with total coolant-flow ratio for three turbine-inlet temperatures. Altitude, 40,000 feet; flight Mach number, 0.788; compressor pressure ratio, 4.42; engine speed, 7600 rpm; stator coolant-flow ratio, 0.01.

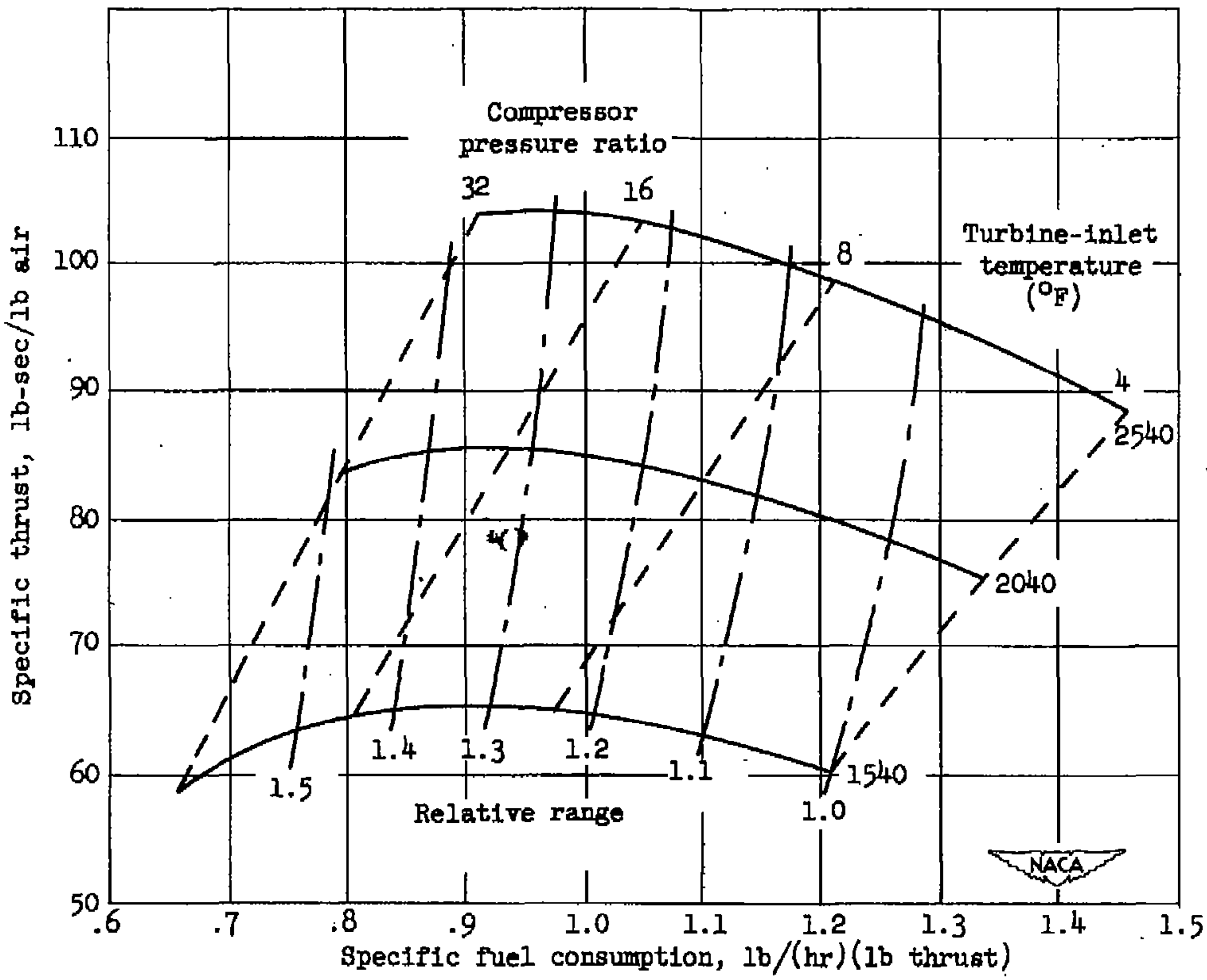


Figure 9. - Performance of turbojet-engine cycle. Flight Mach number, 0.738; altitude, 30,000 feet; compressor and turbine efficiencies, 0.90. (Data from reference 1.)